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IGNITION TRANSIENT ANALYSIS OF SOLID ROCKET MOTOR

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INTRODUCTION

Measurement data on the performance of space shuttle solid rocket motors show wide variations in the head-end pressure changes and the total thrust build-up during the ignition transient periods. To analyse the flow and thermal behavior in the tested solid rocket motors, a 1-dimensional, ideal gas flow model via SIMPLE algorithm was developed [4]. Numerical results showed that burning patterns in the star-shaped head-end segment of the propellent and the erosive burning rate are two important factors controlling the ignition transients.

The objective of the present study is to extend the model to include the effects of aluminum particle commonly used in solid propellents. To treat the effects of aluminum-oxide particles in the combustion gas, conservation of mass, momentum and energy equations for the particles are added in the numerical formulation and integrated by following IPSA [5] approach.

METHOD OF ANALYSIS

Governing Equations

Conservation of mass is

$$\frac{\partial}{\partial t}(r_i \rho_i A) + \frac{\partial}{\partial x}(r_i \rho_i u_i A) = (\omega_i \rho_{PR} - r_i \rho_i)\, r b + \dot{m}_{ig} , \tag{1}$$

where subscript i indicates i component, A is the cross-sectional area of the port, r_i is the volume fraction , ρ_i is the material density, u_i is the velocity, ω_i is the mass fraction in the propellent, ρ_{PR} is the density of the propellent, b is the perimeter of the burning cross-section, $\dot{m}_{ig}$ is the igniter gas flow rate. Burning rate of the propellent is given by [2]

$$r = r_{ref}\,(P/P_{ref})^{n} + \alpha_e G^{0.8} D_n^{-0.2} \exp(-\beta_e r \rho_{PR} / G) . \tag{2}$$

The first part in the RHS of eqn (2) is the standard burning rate and the second part the erosive burning rate. Conservation of mass also requires that

$$\sum r_i = 1.0 \quad ; \quad \sum \omega_i = 1.0 . \tag{3}$$

Conservation of linear momentum is

$$\frac{\partial}{\partial t}(r_i \rho_i u_i A) + \frac{\partial}{\partial x}\left[A\left(r_i \rho_i u_i^2 - r_i \mu_i \frac{\partial u_i}{\partial x}\right)\right] = -r_i A \frac{\partial P}{\partial x} \tag{4}$$

$$- \tau_{wi} P_w - r_i \rho_i r b u_i + A F_{ij} .$$

Thermodynamic pressure of the system is due to gas phase only and is given by

$$P = \rho_1 R_1 T_1 f(\rho_1) , \tag{5}$$

where $f(\rho_1)$ is a correction needed for high pressure gas [3]. The wall shear stress is given by

$$\tau_{w_1} = \frac{1}{8} f \rho_1 u_1^2 , \tag{6}$$

where f is the friction factor obtained by measurement. The inter-phase friction force is given by [1]

$$F_{21} = (18 r_2/d_2^2)\, \mu_1 (u_2 - u_1)(1 + R_e^{2/3}/6) , \tag{7}$$

where d_2 is the particle diameter and the Reynolds number is defined by

$$R_e = \rho_1 |u_2 - u_1| d_2 / \mu_1 . \tag{8}$$

Energy conservation is ,in terms of internal energy,

$$\frac{\partial}{\partial t}(r_i \rho_i e_i A) + \frac{\partial}{\partial x}\left[A \left(r_i \rho_i e_i u_i - r_i \frac{k_i}{c_{v_i}} \frac{\partial e_i}{\partial x} \right) \right] =$$

$$- r_i p \frac{\partial (u_i A)}{\partial x} + u_i \tau_{w_i} \ell_w + \Phi_i - q_i \ell_w + A Q_{ij} \tag{9}$$

$$+ A F_{ij} u_{ij} + \omega_i \rho_{PR} r b h_{f_i} + \dot{m}_{ig} h_{ig} ,$$

where k_i is thermal conductivity, h_{f_i} is the enthalpy of formation, h_{ig} is the enthalpy of igniter gas, and Φ_i is the dissipation.

Heat transfer from the gas to the solid propellent when the propellent surface temperature is lower than a preset auto-ignition temperature is determined by

$$q_1 = h_c (T_1 - T_s), \tag{10}$$

where h_c is a convective heat transfer coefficient determined by measurement. Inter-phase heat transfer is calculated by [1]

$$Q_{21} = (6 k_1 r_2/d_2^2)(T_2 - T_1)(0.58\, R_e^{0.7} P_r^{0.3}), \tag{11}$$

where

$$P_r = \mu_1 c_{p_1} / k_1 . \tag{12}$$

Heat transfer from the gas to the solid propellent is assumed to be one dimensional transient conduction [4].

<u>Initial and Boundary Conditions</u>

Initially stagnant atmospheric air is filled in the chamber and the nozzle. Transient process begins with the introduction of igniter gas into the chamber at the head-end section. Heat transfer from the gas to the solid propellent raises temperature of the propellent resulting in auto-ignition. Mass and energy released from the burning solid propellent rapidly increase momentum and energy of the flow in the system and the total thrust. Open boundary conditions are maintained at the exit of the nozzle and solid wall conditions are used at the head-end.

Numerical Method

IPSA (Inter-Phase-Slip-Algorithm) originated by Spalding [5] is used to approximate the solution of governing equations. IPSA is an extension of SIMPLE method and consequently follows a similar computational procedures. Equation (1) is solved for the volume fraction of particle phase and equation (3) is used to find volume fraction of the gas phase. With known volume fractions and a guessed pressure field, momentum equations are solved for velocities. Corrections on density, velocity and pressure are made by solving pressure correction equation. Energy equation is solved to update temperature change. These steps are repeated until convergence is satisfied within a given time step.

RESULTS AND CONCLUSIONS

Geometry of the motor, igniter gas flow rate and other physical and numerical parameters are those used in a previous analysis based on a single fluid model [4].

Numerical results obtaind by the single fluid option of the present model are shown in Figure 1 in comparison with measurement data. Numerically predicted pressure increase after the flame spreding (0.12 sec) is higher than the measured data. Higher pressure increase is due to higher burning rate which is caused by higher flow speed in the chamber. Predicted total thrust is much higher than the measured value (about 46 % higher at 0.6 sec).

Numerical results using two-fluid option of the model are shown in Figure 2. Particle diameter is assumed to be 10 m and the density of the particle is 1500 kg/m . Particle loading of the aluminum in the solid propellent is 10 %. Due to increased mixture density, flow speed in the chamber is much slower and subsequently slower increases in pressure gradient and total thrust. Numerical results are in agreement with physical expectations.

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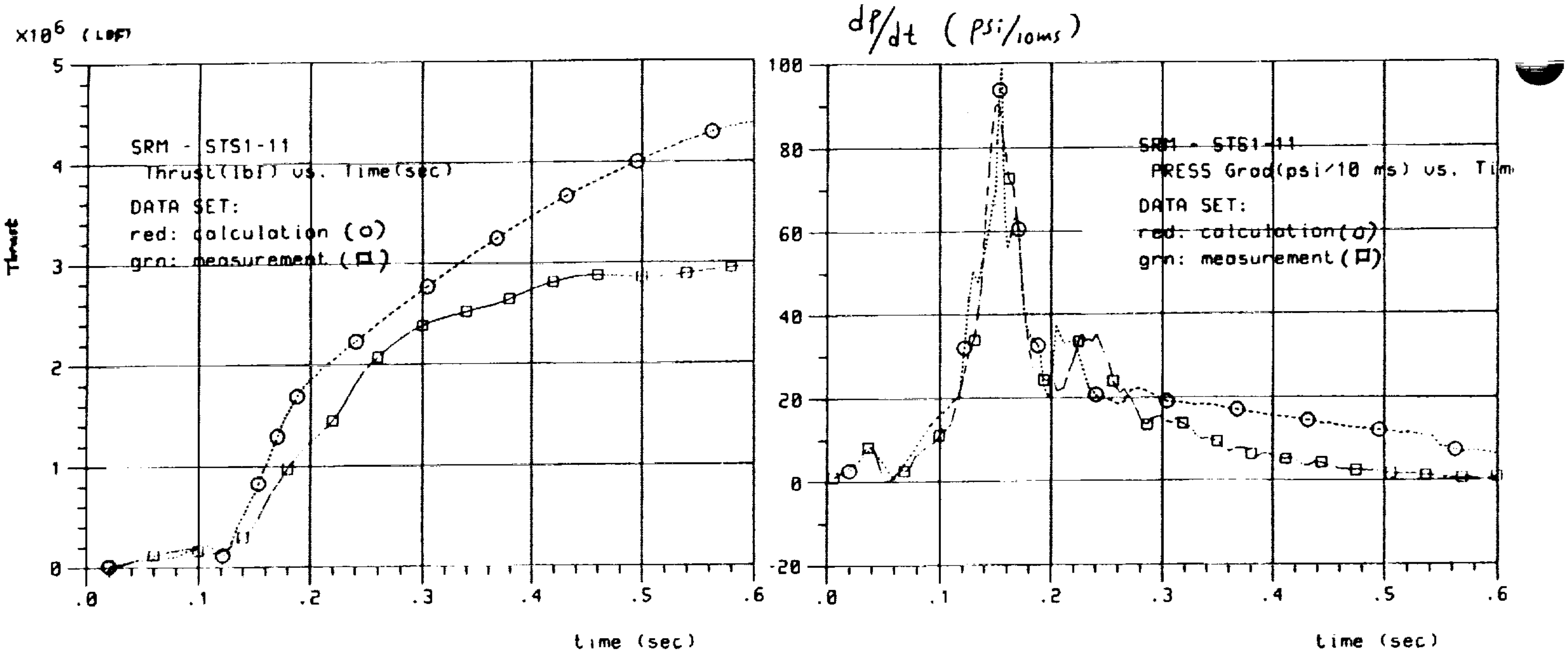
x10^6 (LBF)
Thrust
SRM - STS1-11
Thrust(lbf) vs. Time(sec)
DATA SET:
red: calculation (o)
grn: measurement (□)
time (sec)
dP/dt (psi/10ms)
SRM - STS1-11
PRESS Grad(psi/10 ms) vs. Tim
DATA SET:
red: calculation(o)
grn: measurement (□)
time (sec)

Fig. 1

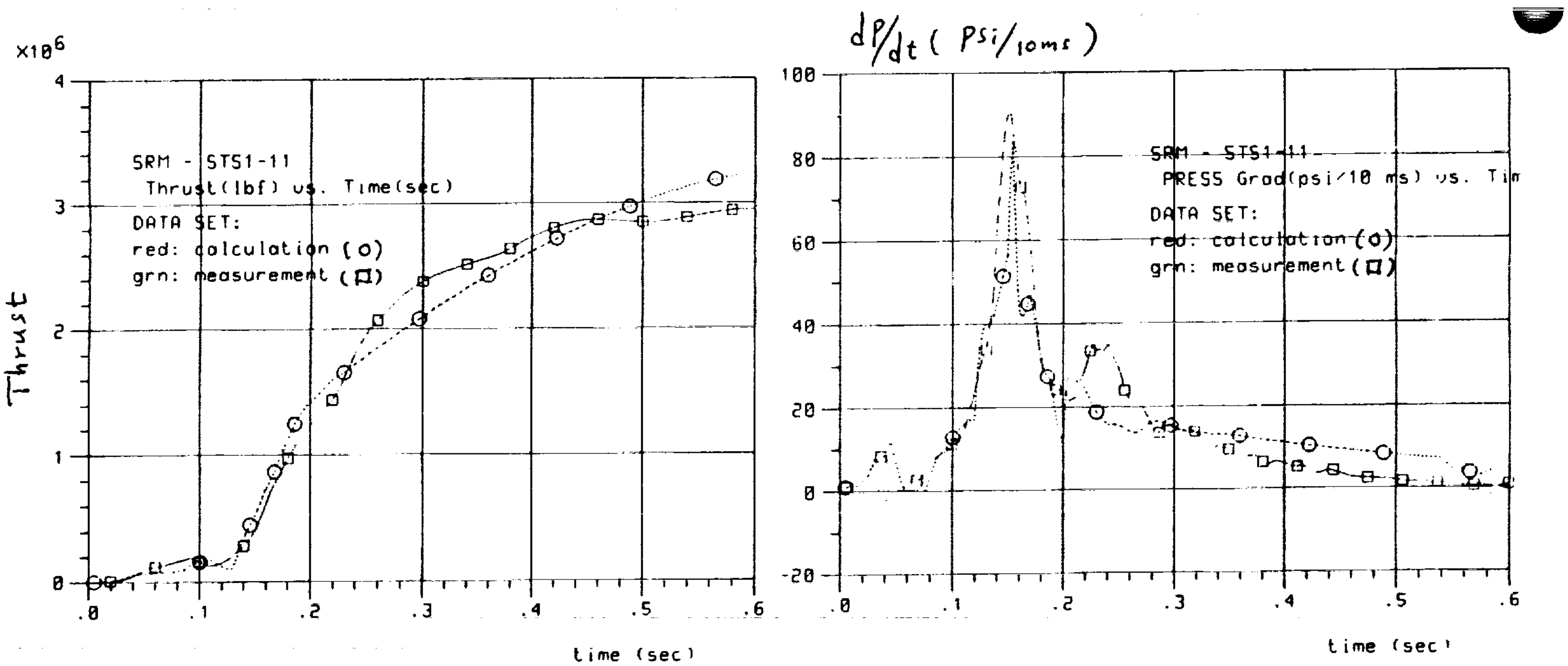
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Fig. 2